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An Analysis of Athena Tracking Problems (The Athena Portion of ABRES 627a)

18 NOVEMBER 1963

Prepared by LAWRENCE NORWOOD
Telecommunications and Tracking Department

Prepared for COMMANDER HEADQUARTERS BALLISTIC SYSTEMS DIVISION

AIR FORCE SYSTEMS COMMAND

UNITED STATES AIR FORCE

Norton Air Force Base, California



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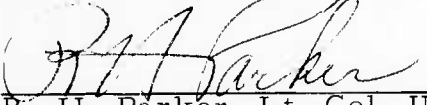
AN ANALYSIS OF ATHENA TRACKING PROBLEMS
(THE ATHENA PORTION OF ABRES 627a)

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ABSTRACT

The ATHENA program has been conceived to provide a systematic method for investigation of atmospheric reentry phenomena and, through the use of a test vehicle especially designed to reproduce ICBM trajectory dynamics, to facilitate the evaluation of advanced reentry concepts. The White Sands Missile Range has been selected to make critical reentry measurements. Green River, Utah, will be the launch site.

Tracking radars will be provided at the launch site to perform flight safety surveillance, to provide acquisition data for range radars, and to monitor performance during the initial portion of the trajectory.

This report contains discussions on (1) launch radar tracking precision, (2) launch radar optimum placement, (3) radar system and missile transponder parameters, and (4) maximum radar and missile transponder tracking capability.

CONTENTS

1.0	INTRODUCTION	1
2.0	LAUNCH RADAR TRACKING ACCURACY REQUIREMENTS	3
3.0	LAUNCH AREA RADAR SITING	15
3.1	Geometrical Relationships	16
3.2	The Beacon Equation	18
3.3	Conclusions	27
4.0	TRACKING LINK CALCULATIONS	29
4.1	Second-Stage Beacon Tracking	29
4.1.1	Launch Site to Termination of Second Thrust	29
4.1.2	Either Site to Maximum Range	29
4.2	Fourth Stage Beacon Tracking - Maximum Range. .	31
4.3	Radar (Skin) Tracking	31
4.3.1	Launch Site to Termination of Second Thrust	31
4.3.2	Either Site to Maximum Range	34

1.0 INTRODUCTION

The ATHENA program has been conceived to provide a systematic method for the investigation of atmospheric reentry phenomena and, through the use of a test vehicle especially designed to reproduce ICBM trajectory dynamics, to facilitate the evaluation of advanced reentry concepts.

The White Sands Missile Range has been selected to make critical reentry measurements, the selection having been made on the bases of existing instrumentation and the anticipated ease of satisfaction of flight test requirements. Among these requirements, that of flight safety surveillance is pre-eminent.

Tracking radars will be provided at the launch site to perform flight safety surveillance, to provide acquisition data for range radars and to monitor performance during the initial portion of the trajectory. Of the several questions which pertain to tracking performance, the following are discussed in this analysis.

- a. With what precision must the launch radars track, in order to enable the range radars to effect acquisition?
- b. What is the optimum placement of the launch radars?
- c. What radar system parameters and what missile transponder parameters are necessary to insure satisfactory tracking performance, at least to that critical point of the trajectory at which second stage thrust is terminated?
- d. Having determined these parameters, what are the maximum ranges to which the missile can be tracked, both in the radar (skin) tracking mode and in the beacon (transponder) mode?

Among the questions which remain for resolution, the following are probably most important.

- a. What radar resolution may be required to distinguish between the third and fourth stages, after third stage separation?

Air Force Report No.
BSD-TDR-63-144

Report No.
TDR-169(3153-01)TN-6

- b. What techniques of discrimination should be employed to insure payload tracking after fourth stage separation? More particularly, will the payload be hidden by the flame plasma of the fourth stage retrorocket? Or will this flame plasma by its persistence constitute background clutter and partially obscure the payload?
- c. To what extent will the hypersonic reentry plasma sheath interfere with tracking capability?

2.0 LAUNCH RADAR TRACKING ACCURACY REQUIREMENTS

The FPS-16 launch radars will track the target in the beacon mode, and target positions will be specified by the spherical coordinates (r_1, θ_1, ϕ_1) , referred to the launch radar. These coordinates will be transmitted to a computer which will perform a coordinate transformation and compute the target position in the spherical coordinates (r_2, θ_2, ϕ_2) , referred to the range radar. The problem, then, is to determine the effect of errors in (r_1, θ_1, ϕ_1) , upon (r_2, θ_2, ϕ_2) . The assumptions are as follows:

- a. Geodetic errors in radar survey are negligible.
- b. The difference in local radar altitudes will be accommodated by the computer.
- c. Data link transmission errors are not considered.
- d. Computer errors are not considered.

The launch radar and range radar, together with the geocenter, determine a plane which intersects the earth in a great circle. In general, the slant ranges, r_1 and r_2 , are not in this plane. In the ATHENA geometry, however, the angles which the slant ranges make with the great circle plane will be small, so it is convenient to replace the actual slant ranges by their projections in the great circle plane. (This is equivalent to assuming that both radars are in the plane of the trajectory, which reduces the immediate problem to two dimensions and permits us to consider azimuth errors separately.)

In Figure 1, we let a be the earth's radius and α be the geocentric angle. Two perpendicular constructions are required. The first is from the range radar to the earth's radius at the launch radar. The second is from the target to the first perpendicular.

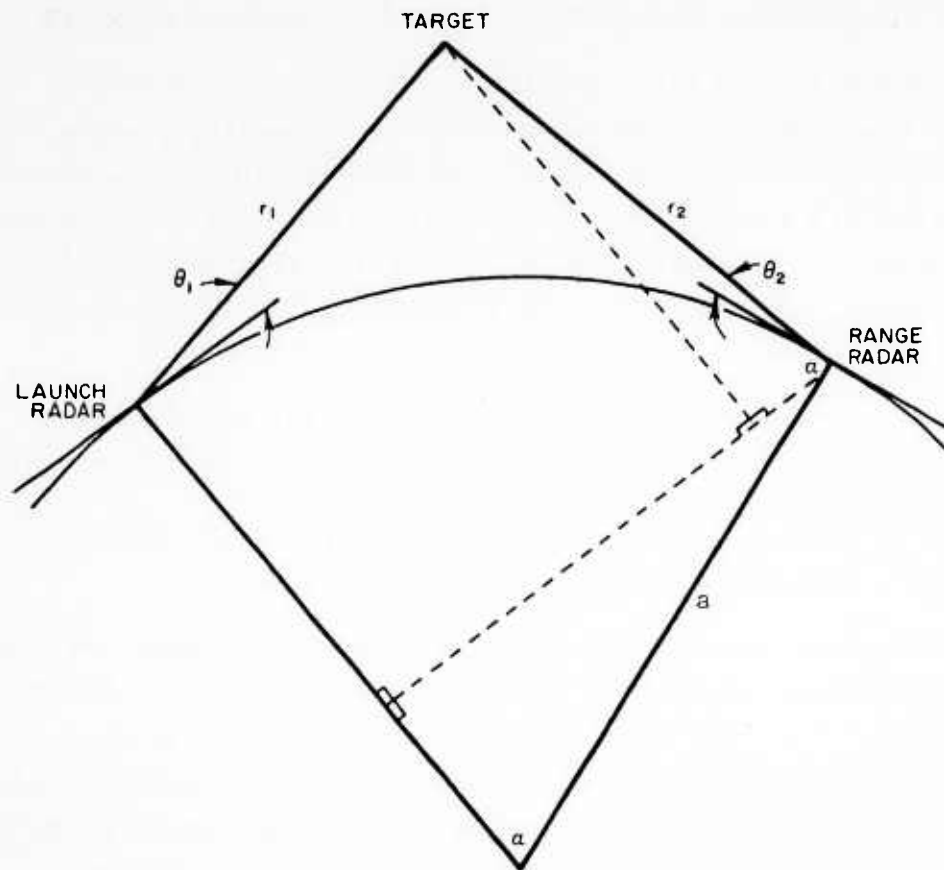


Figure 1. Coordinate Transformations

The Cartesian coordinates of the target (x_1, y_1) , referred to the launch radar, are related to the spherical coordinates of the range radar by

$$\begin{aligned} x_1 &= a \sin \alpha - r_2 \cos (\theta_2 + \alpha) \\ y_1 &= r_2 \sin (\theta_2 + \alpha) - a + a \cos \alpha \end{aligned} \tag{1}$$

whence

$$\begin{aligned} dx_1 &= r_2 \sin(\theta_2 + \alpha) d\theta_2 - \cos(\theta_2 + \alpha) dr_2 \\ dy_1 &= r_2 \cos(\theta_2 + \alpha) d\theta_2 + \sin(\theta_2 + \alpha) dr_2 \end{aligned} \quad (2)$$

Also,

$$\begin{aligned} x_1 &= r_1 \cos \theta_1 \\ y_1 &= r_1 \sin \theta_1 \end{aligned} \quad (3)$$

whence

$$\begin{aligned} dx_1 &= \cos \theta_1 dr_1 - r_1 \sin \theta_1 d\theta_1 \\ dy_1 &= \sin \theta_1 dr_1 + r_1 \cos \theta_1 d\theta_1 \end{aligned} \quad (4)$$

so that from (4):

$$\begin{aligned} dr_1 &= \sin \theta_1 dy_1 + \cos \theta_1 dx_1 \\ r_1 d\theta_1 &= \cos \theta_1 dy_1 - \sin \theta_1 dx_1 \end{aligned} \quad (5)$$

and from (2) and (5):

$$\begin{aligned} dr_1 &= [\sin \theta_1 \sin (\theta_2 + \alpha) - \cos \theta_1 \cos (\theta_2 + \alpha)]dr_2 \\ &\quad + r_2[\cos \theta_1 \sin (\theta_2 + \alpha) + \sin \theta_1 \cos (\theta_2 + \alpha)]d\theta_2 \\ r_1 d\theta_1 &= [\sin \theta_1 \cos (\theta_2 + \alpha) + \cos \theta_1 \sin (\theta_2 + \alpha)]dr_2 \\ &\quad + r_2[\cos \theta_1 \cos (\theta_2 + \alpha) - \sin \theta_1 \sin (\theta_2 + \alpha)]d\theta_2 \end{aligned} \quad (6)$$

The range radar is required to establish automatic tracking before the target reaches an altitude (relative to the horizon plane of the launch radar) of 450,000 feet*, for the trajectory of lowest re-entry angle (18°).

For the trajectory of lowest re-entry angle, the target will reach an altitude of $y_1 = 450,000$ feet at a point down range from the launch radar a distance of $x_1 = 510,000$ feet (Reference 1), so that $r_1 = 680,150$ feet and $\theta_1 = \sin^{-1} 0.66162 = \cos^{-1} 0.74983$.

The launch radar and range radar are separated by a maximum of 410 miles of arc along the great circle, so the geocentric angle is given by

$$\alpha = \frac{410}{a} \text{ radians} = \frac{(410)(180)}{\pi a} \text{ degrees.}$$

Since

$$a = 3959 \text{ miles,} \quad \alpha = 5.93^\circ$$

*It is worth noting that for any "earlier" point on the trajectory, the target coordinates with respect to the launch radar will be known with greater precision, so that less accuracy will be required of the range radar.

Thus

$$\sin \alpha = 0.1033$$

$$\cos \alpha = 0.9946$$

and equations (1) become

$$510,000 = (3959)(5280)(0.1033) - r_2 \cos(\theta_2 + \alpha)$$

$$450,000 = r_2 \sin(\theta_2 + \alpha) - (3959)(5280)(1 - 0.9946)$$

or

$$r_2 \cos(\theta_2 + \alpha) = 1,649,334$$

$$r_2 \sin(\theta_2 + \alpha) = 562,879$$

from which

$$\tan(\theta_2 + \alpha) = 0.3413$$

$$\sin(\theta_2 + \alpha) = 0.3230$$

$$\cos(\theta_2 + \alpha) = 0.9464$$

$$r_2 = 1,742,700$$

Equations (6) then become

$$dr_1 = -0.4959dr_2 + 0.8683r_2d\theta_2 \quad (7)$$

$$r_1d\theta_1 = 0.8683dr_2 + 0.4959r_2d\theta_2$$

or

$$dr_1 = -0.4959dr_2 + 1,513,186d\theta_2 \quad (8)$$

$$d\theta_1 = 1.2766 \cdot 10^{-6}dr_2 + 1.2706d\theta_2$$

where range is in feet and angle is in radians.

The range radar is an AN/FPS-16, for which the accuracy requirements for automatic acquisition are that range must be known within 2400 feet and that angle must be known within 0.5 degrees. If we let $dr_2 = 2400$ and $d\theta_2 = 0.5$ degrees = 0.0087 radians in Equations (8),

$$dr_1 = 11,975 \text{ feet and } d\theta_1 = 0.0151 \text{ radians} = 15.1 \text{ milliradians.}$$

These values of dr_1 and $d\theta_1$ obviously represent necessary conditions for launch radar tracking accuracy. One might infer erroneously that these values are also sufficient. That this reasoning is specious will be demonstrated by transforming the coordinates in the other order.

In this case, the two constructions of Figure 1 are a perpendicular from the launch radar to the earth's radius at the range radar, and a second perpendicular from the target to the first perpendicular. Equations (1) through (6) are then unchanged, except for an interchange of subscripts throughout. The remainder of the argument then proceeds in the following logical order:

$$\begin{aligned}x_1 &= 510,000 \\y_1 &= 450,000 \\r_1 &= 680,150 \\\sin \theta_1 &= 0.6616 \\\cos \theta_1 &= 0.7498 \\\theta_1 &= 41.42^\circ \\a &= 5.93^\circ \\\sin a &= 0.1033 \\\cos a &= 0.9946 \\\theta_1 + a &= 47.35^\circ \\\sin (\theta_1 + a) &= 0.7355 \\\cos (\theta_1 + a) &= 0.6776\end{aligned}$$

Equations (1), with interchanged subscripts, then become

$$x_2 = (3959)(5280)(0.1033) - 680,150(0.6766) = 1,699,150$$

$$y_2 = 680,150(0.7355) - (3959)(5280)(1 - 0.9946) = 387,371$$

From Equations (1), then, we know that

$$r_2 = 1,742,700$$

$$\sin \theta_2 = 0.2223$$

$$\cos \theta_2 = 0.9750$$

We then turn to Equations (6) and interchange subscripts to obtain Equations (6'), which become, after appropriate substitution,

$$dr_2 = -0.4972dr_1 + 0.8677r_1d\theta_1 \quad (7')$$

$$r_2d\theta_2 = 0.8677dr_1 + 0.4972r_1d\theta_1$$

We note the close similarity between equations (7) and (7'), which is to be expected. When, in Equations (7') we substitute for r_1 and r_2 to obtain Equations (8'),

$$dr_2 = 0.4972dr_1 + 590,195d\theta_1 \quad (8')$$

$$d\theta_2 = 0.4979 \cdot 10^{-6}dr_1 + 0.1940d\theta_1$$

We note, in comparing Equations (8') with Equations (8), the dissimilarity and thus the significance of target range.

If we substitute into Equations (8') the previous "apparent" requirements for launch radar tracking accuracy, $dr_1 = +11,975$ and $d\theta_1 = +0.0151$ radians, we find,

$$\begin{aligned} dr_2 &= -0.4972(11,975) + 590,195(0.0151) \\ &= -5,954 + 8,972 = 2958 \text{ feet} \\ d\theta_2 &= 0.4979 \cdot 10^{-6}(11,975) + 0.1940(0.0151) \\ &= 0.00596 + 0.00293 = 0.0089 \text{ radians} \end{aligned}$$

These values of dr_2 and $d\theta_2$ are not surprising and, indeed, are almost identical with the accuracy requirements for range radar automatic acquisition, as we would expect. However, it is now clear that the "apparent" values of $dr_1 = 11,975$ and $d\theta_1 = 0.0151$ are intolerably large. In fact, what we have shown is that it is necessary to consider the launch radar tracking errors, dr_1 and $d\theta_1$, as positive or negative, and to treat them in Equations (8'), as additive.

We could, for example, determine the maximum tracking errors permissible in the launch radar by a max-min procedure, as follows:

$$\begin{aligned} \max dr_1 &= \min[0.4972dr_1 = 2400; 0.4979 \cdot 10^{-6}dr_1 = 0.0087] \\ &= \min[dr_1 = 4827; dr_1 = 17,473] = 4827 \end{aligned}$$

Note that this value of dr_1 would permit no error in θ_1 .

$$\begin{aligned} \max d\theta_1 &= \min[590,195d\theta_1 = 2400; 0.1940d\theta_1 = 0.0087] \\ &= \min[d\theta_1 = 0.0041; d\theta_1 = 0.0448] = 0.0041 \end{aligned}$$

Note that this value of $d\theta_1$ would permit no error in r_1 . In particular, this excludes the AN/SCR-584 as the launch radar, since its reliable angular accuracy is of the order of 5 to 6 milliradians (Reference 2).

A more conservative approach is to employ a minimax procedure in the interpretation of Equations (8'). Thus,

$$1/2dr_1 < \max[0.4972dr_1, 590, 195d\theta_1] < 1200$$

$$1/2d\theta_1 < \max[0.4979 \cdot 10^{-6}dr_1, 0.1940d\theta_1] < 0.00435$$

whence

$$dr_1 < \min[0.4972dr_1 = 1200; 0.4979 \cdot 10^{-6}dr_1 = 0.00435]$$

$$< \min[dr_1 = 2413; dr_1 = 8736] = 2413$$

$$d\theta_1 < \min[590, 195d\theta_1 = 1200; 0.1940d\theta_1 = 0.00435]$$

$$< \min[d\theta_1 = 0.00203; d\theta_1 = 0.0224] = 0.00203$$

Note that these launch radar inaccuracies, $dr_1 \simeq 2400$ feet and $d\theta_1 \simeq 2$ milliradians, can be tolerated simultaneously.

If we let $d\theta_1 = 1$ milliradian in Equations 8', we find the maximum value of dr_1 which can be tolerated simultaneously to be $dr_1 = 3640$.

If we let $d\theta_1 = 3$ milliradians, the maximum tolerable value of dr_1 is 1266.

If we let $d\theta_1 = 4$ milliradians, the maximum tolerable value of dr_1 is 78.

These pairs of values for $d\theta_1$ and dr_1 are summarized in Table 1 and Figure 2.

Table 1. Maximum Tolerable Launch Radar Tracking Errors

$d\theta_1$ MILLIRADIANS	dr_1 FEET
0	4827
1	3640
2	2400
3	1266
4	78
4.1	0

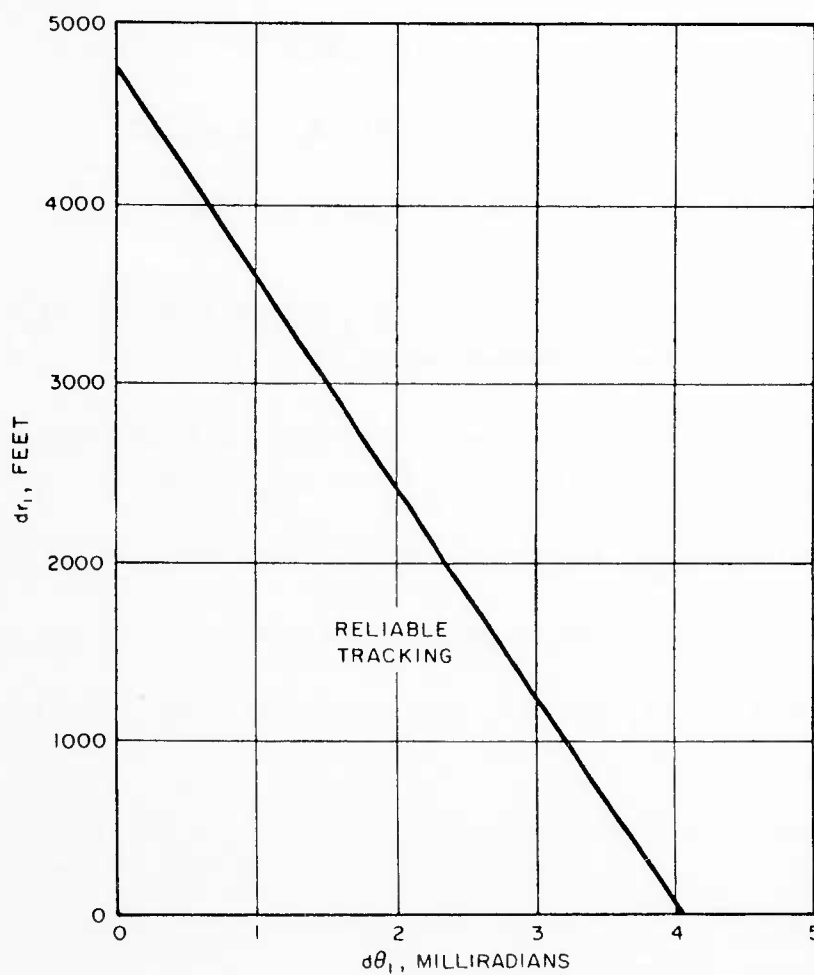


Figure 2. Launch Radar Tracking Accuracy Requirements

Since azimuth error is defined in the horizon plane,

$$x_1 d\phi_1 = x_2 d\phi_2$$

$$d\phi_2 = \frac{x_1}{x_2} d\phi_1 = \frac{510,000}{1,699,150} d\phi_1 = 0.3 d\phi_1$$

which says that the range radar azimuth error will be less than one-third the launch radar azimuth error for the assumed vehicle location.

It should be noted that References 3 and 4 state the launch radar accuracy requirements to be ± 66 feet in range and ± 4 milliradians in angle. These values fall in the "reliable" area, under the straight line of Figure 2, but just barely. It should be evident from the preceding discussion that it would be preferable not to impose such values as requirements.

In any case, the conclusions are that the AN/MPS-26 would be more than adequate (with a range accuracy of ± 30 feet and an angular accuracy of ± 1.5 milliradians) and, of course, that the AN/FPS-16 would be still better (with a range accuracy of ± 30 feet and an angular accuracy of ± 0.5 milliradians). Cf. Reference 2.

3.0 LAUNCH AREA RADAR SITING

Earlier memoranda (References 5-9) have expressed concern over the possibility of loss of C band beacon tracking capability during the later portion of second stage propulsion, if the tracking antenna were to be located in the plane of the trajectory and in the immediate vicinity of the launch pad.

That this concern is realistic is supported by reports of Polaris tracking performance (References 10-11), which indicate strong dependence of flame attenuation upon aspect angle, and very severe attenuation for tail aspect tracking.

Discussions at AMR (Reference 12) produced agreement among representatives of Aerospace, AMR, and WSMR, that a comparative evaluation of three proposed sites be made on the basis of anticipated signal margins (or, stated a little differently, that for each of the three sites, the requirements for beacon transmitter power and antenna directivity be determined).

The purpose of this paragraph is to present qualitatively such an evaluation, and to document subsequent agreements made at WSMR during the conference of Reference 13.

Paragraph 3.1 depicts the essential geometrical relationships, and directs attention to that critical point of the trajectory at which second stage burnout will occur.

Paragraph 3.2 states the beacon equation, and presents detailed calculations of signal margin for each of the three proposed sites.

Paragraph 3.3 summarizes the discussions and presents the consensus of the meeting of Reference 13.

3.1 GEOMETRICAL RELATIONSHIPS

Figure 3 depicts the plane of the trajectory with launch pad at L, and indicates that second stage burnout will occur at P, at an altitude of 40 miles and at a distance 25 miles downrange from L.

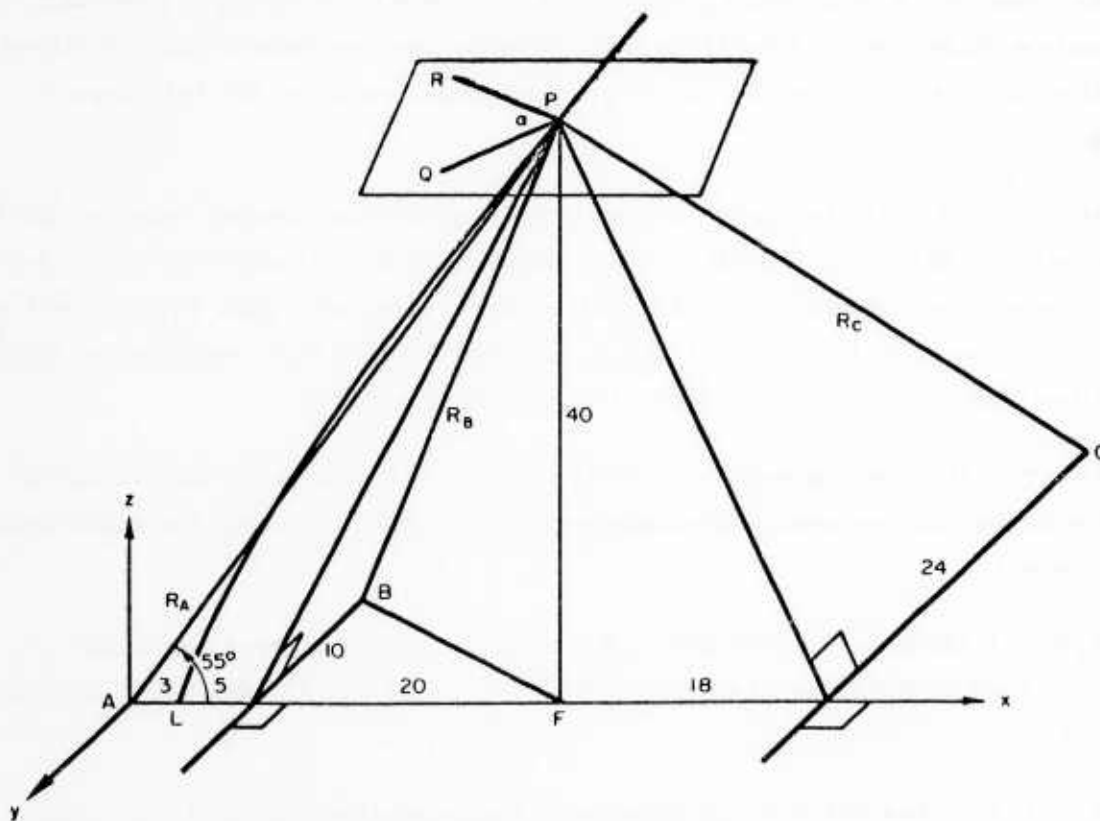


Figure 3. Launch Area Geometry

Site A is in the plane of the trajectory and 3 miles behind the launch pad. The beacon range (to P) is given by

$$R_A^2 = 40^2 + 28^2 = 2384 \text{ mi}^2$$

Site B is 5 miles down-range and 10 miles cross-range. The beacon range is given by

$$R_B^2 = 40^2 + 20^2 + 10^2 = 2100 \text{ mi}^2$$

Site C is 43 miles down-range and 24 miles cross-range. The beacon range is given by

$$R_C^2 = 40^2 + 18^2 + 24^2 = 2500 \text{ mi}^2$$

At P (second stage burnout) it is known that the flight path angle will be 55° (for the least favorable case, in which the elevation angle at launch is 78° and the re-entry angle is 42.8°). We note that this flight path angle is identical with the elevation angle of the radar at site A, which is given by

$$\text{angle PAL} = \tan^{-1} \frac{40}{28} = \tan^{-1} 1.43 = 55^\circ$$

Thus, the radar aspect angle at site A is 180° (measured from the nose).

At site B, the radar aspect angle is given by the supplement of angle APB, which latter can be determined from angle APB by the law of cosines.

$$\begin{aligned} \text{angle APB} &= \cos^{-1} \frac{R_A^2 + R_B^2 - (AB)^2}{2R_A R_B} = \cos^{-1} \frac{2384 + 2100 - (8^2 + 10^2)}{2\sqrt{2384} \cdot 2100} \\ &= \cos^{-1} \frac{54}{\sqrt{3129}} = \cos^{-1} 0.9654 \approx 15^\circ \end{aligned}$$

Thus, the radar aspect angle at site B is $\sim 165^\circ$.

At site C, the radar aspect angle is given by the supplement of angle APC;

$$\begin{aligned} \text{angle APC} &= \cos^{-1} \frac{R_A^2 + R_C^2 - (AC)^2}{2R_A R_C} = \cos^{-1} \frac{2384 + 2500 - (46^2 + 24^2)}{2\sqrt{2384 \cdot 2500}} \\ &= \cos^{-1} \frac{137}{25\sqrt{149}} = \cos^{-1} 0.4489 \approx 63^\circ \end{aligned}$$

Thus, the radar aspect angle at site C is $\sim 117^\circ$.

3.2 THE BEACON EQUATION

We write the beacon equation in the form:

$$P_{R_a} = \frac{P_T G_T G_R}{L_T L_P L_R}, \quad (1)$$

where

P_{R_a} = received power,

P_T = transmitted power,

L_T = transmitter losses (transmission line),

G_T = transmitter antenna directivity,

L_P = path loss (free space loss plus flame attenuation, where applicable),

G_R = receiver antenna directivity,

L_R = receiver losses (transmission line and diplexer).

The actual received power, P_{R_a} , is given by equation (1), whereas the required signal power, P_{R_r} , is given by

$$P_{R_r} = kTB S/N \quad (2)$$

(where T is the effective receiver system noise temperature), so that the signal margin is given by $P_{R_a} - P_{R_r}$.

(To facilitate algebraic addition, link parameters are written as ten times their logarithm to the base 10).

We assume that the transmitter power output will be 600 watts, so that $P_T = 27.8$ dbw. We assume that the transmitter losses will not exceed $L_T = 1.5$ db.

Figure 4 depicts the radiation pattern of the Azusa beacon antenna for the Polaris A2X vehicle (which is roll-stabilized). Although considerable improvement upon this pattern can be realized for ATHENA requirements, it is convenient for this computation to regard Figure 4 as typical of a C band slot antenna.

Since the ATHENA vehicle will not be roll-stabilized, we must concern ourselves with the antenna obscuration time. The specified ATHENA spin rate is 3 rps, which means that the vehicle will make one (roll) revolution in $1/3$ second. If we assume that the maximum tolerable signal drop out period is $1/4$ second, this implies that we must require a detectable signal for $(1/3 - 1/4 =) 1/12$ second during each revolution (i.e., as ϕ , the vehicle reference angle of Figure 4, varies $\pm 45^\circ$ from the direction of maximum radiation intensity). Figure 4 then indicates that the maximum antenna directivity which will insure no more than $1/4$ second (or 75%) obscuration is

-6 db for 180° aspect (site A),
1 db for 165° aspect (site B),
4 db for 117° aspect (site C).

These values for gain are correct only if we assume continuous polarization alignment (which automatically will be the case for site A, since it is in the plane of the trajectory). For sites B and C (since the FPS-16 does not track polarization) we must determine the polarization mis-alignment angle, α , depicted in Figure 3, and adjust the gain by a factor of $\cos^2 \alpha$.

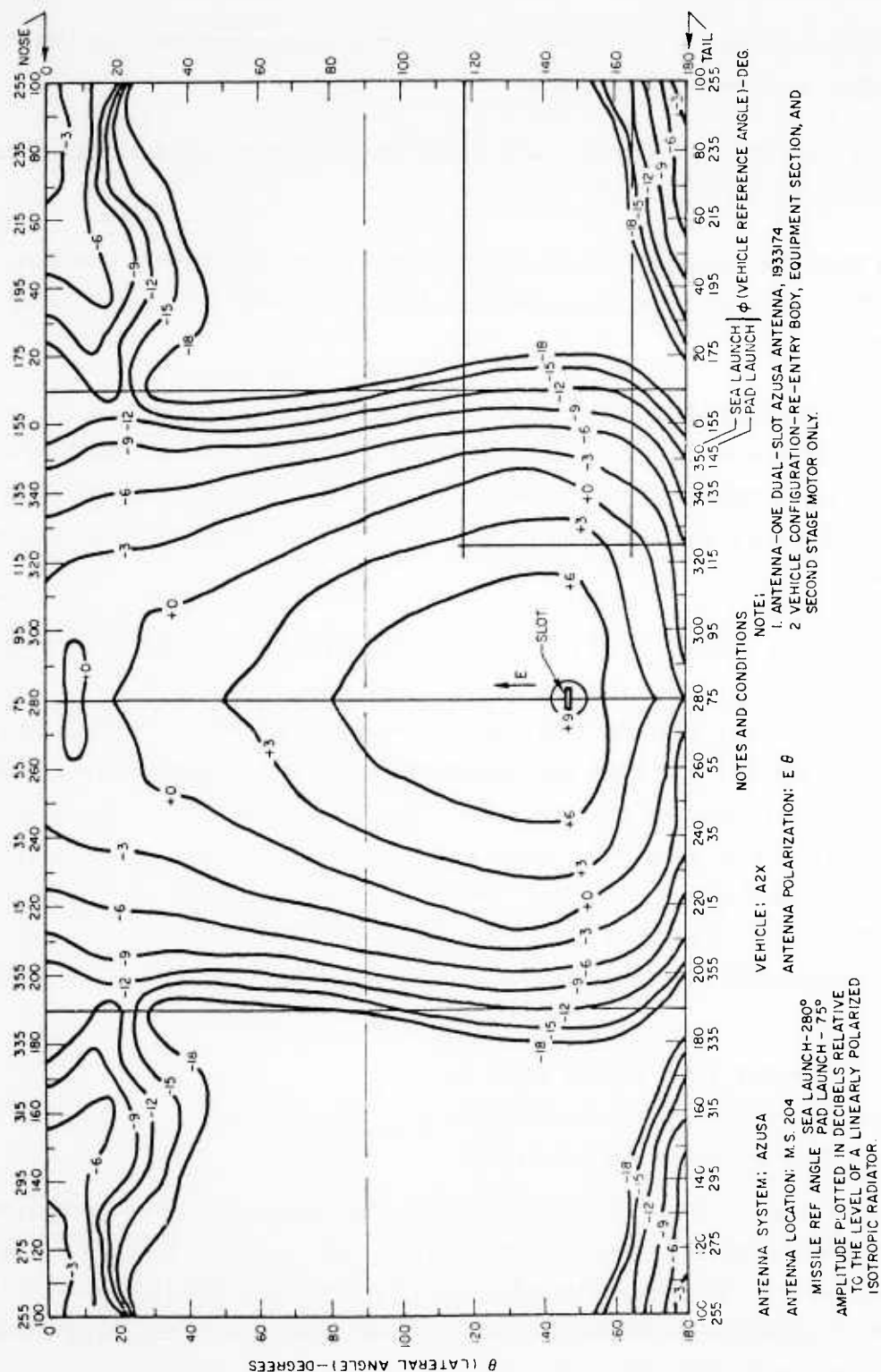


Figure 4. Antenna Radiation Pattern

In Figure 3, we project the missile axis, AP, on a plane perpendicular to the radar line of sight, BP, and denote the projection by PQ. Thus PQ, PA and PB are coplanar. If the polarization at B is vertical, its direction is perpendicular to BP and lies in the plane BPF. We draw this polarization vector coplanar with PQ, and denote it by PR. Thus, the polarization misalignment angle is given by $\alpha = \text{angle QPR}$, and we note that α is also the dihedral angle between the planes APB and BPF.

We superimpose a Cartesian coordinate system upon Figure 3, with the origin at A, the x-axis along ALF, the z-axis in the plane of the trajectory and in the direction of FP, and the y-axis in the direction away from B.

The plane APB passes through A (0,0,0), P (28,0,40), and B (8,-10,0), so its equation is $10x + 8y - 7z = 0$.

The plane BPF passes through B (8,-10,0), P (28,0,40), and F (28,0,0), so its equation is $x - 2y - 28 = 0$.

The acute angle between the two planes is given by

$$\cos \alpha_B = \frac{10 - 16 + 196}{\sqrt{100 + 64 + 49} \cdot \sqrt{1 + 4 + 784}} = \frac{190}{\sqrt{168057}}$$

$$\cos^2 \alpha_B = \frac{36100}{168057} = 0.215 \quad (\text{and } \alpha_B \approx 62^\circ 23')$$

The adjusted beacon antenna directivity, for site B, therefore would be

$$(1 \text{ db})(0.215) = (1.26)(0.215) = 0.271 = -5.7 \text{ db} ,$$

which indicates that considerably less than half the radiated signal energy will be received by a vertically polarized antenna at site B. Thus, we assume that the tracking antenna at site B, although beginning track from launch with vertical polarization, will switch to horizontal polarization before the missile reaches P.

The adjusted beacon antenna directivity (with horizontal polarization) then is

$$G_T = (1 \text{ db}) \sin^2 \alpha_B = (1.26)(0.785) = 0.989 = 0 \text{ db} ,$$

(to the nearest tenth) for site B.

In a similar fashion, for site C, the plane of APC passes through A (0, 0, 0), P (28, 0, 40), and C (46, -24, 0), so its equation is $60x + 115y - 42z = 0$.

The plane CPF passes through C (46, -24, 0), P (28, 0, 40), and F (28, 0, 0), so its equation is $4x + 3y - 112 = 0$.

The acute angle between the two planes is given by

$$\cos \alpha_C = \frac{240 + 345 + 4704}{\sqrt{3600 + 13225 + 1764} \sqrt{16 + 9 + 12544}} = \frac{5289}{\sqrt{233645141}}$$

$$\cos^2 \alpha_C = \frac{27973521}{233645141} = 0.120 \quad (\text{and } \alpha_C \approx 69^\circ 46')$$

The adjusted beacon antenna directivity, for site C, therefore would be

$$(4 \text{ db})(0.12) = (2.51)(0.12) = 0.301 = -5.2 \text{ db} .$$

which indicates that very much less than half the radiated signal energy will be received by a vertically polarized antenna at site C. Thus, as before, we assume that the tracking antenna at site C, although beginning track from launch with vertical polarization, will switch to horizontal polarization before the missile reaches P.

The adjusted beacon antenna directivity (for horizontal polarization) then is

$$G_T = (4 \text{ db}) (\sin^2 \alpha_C) = (2.51)(0.88) = 2.21 = 3.4 \text{ db} ,$$

for site C.

Summarizing, the transmitter antenna directivity is

$$G_T = -6.0 \text{ db for } 180^\circ \text{ aspect (site A),}$$

$$G_T = 0.0 \text{ db for } 165^\circ \text{ aspect (site B),}$$

$$G_T = 3.4 \text{ db for } 117^\circ \text{ aspect (site C).}$$

The free space loss is given by $\frac{16\pi^2 R^2}{\lambda^2}$, or if we express R in miles and f = frequency in Gc, the free space loss is $R^2 f^2$ (96.6 db). For a carrier frequency of $f = 5.4$ Gc, $f^2 = 29.16 = 14.6$ db.

For site A, $R_A^2 = 2384 = 33.8$ db, and the free space loss is 145.0 db.

For site B, $R_B^2 = 2100 = 33.2$ db, and the free space loss is 144.4 db.

For site C, $R_C^2 = 2500 = 34.0$ db, and the free space loss is 145.2 db.

Reference 7 notes that the ATHENA second stage rocket is similar to the Polaris A2X second stage rocket in chemical composition, exhaust products, specific impulse, etc. For this reason the measured performance of four FPS-16's at AMR in tracking the Polaris A2X is believed to be directly indicative of the anticipated performance in tracking the ATHENA; especially as regards the dependence of flame attenuation of signal energy at C band upon aspect angle.

Figure 5, from Reference 10, depicts the Azusa received signal strength, the dashed curve being the predicted signal level based upon the missile trajectory, the aspect angle, and the antenna pattern. The solid curve shows the received signal level. The difference between the two curves depicts the magnitude of the flame effect, and indicates that the flame attenuation exceeded 40 db, at least for this one flight.

Reference 11 presents the conclusion, from several Polaris tests, that flame attenuation is negligible so long as the aspect angle does not exceed 160° . It is also observed in Reference 11 that C band flame attenuation exhibits a steep

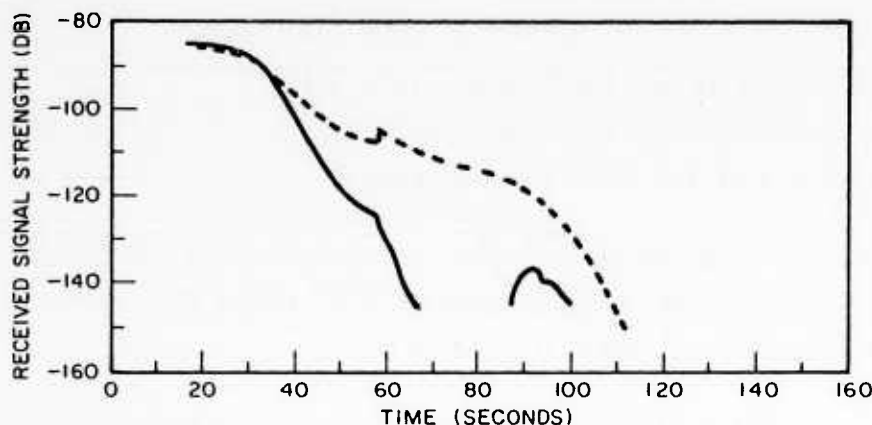


Figure 5. Received Signal Strength for an A2x Polaris Flight

gradient when the aspect angle exceeds 163° . Maximum attenuation values* are noted to have reached 37 db for aspect angles in excess of 173° .

Thus, we conclude that the path loss is

$$\begin{aligned}
 L_p &= \text{free space loss} + \text{flame attenuation} \\
 &= 145.0 \text{ db} + 37.0 \text{ db} = 182.0 \text{ db for site A,} \\
 &= 144.4 \text{ db} + \text{none} = 144.4 \text{ db for site B,} \\
 &= 145.2 \text{ db} + \text{none} = 145.2 \text{ db for site C.}
 \end{aligned}$$

From Reference 2, $G_R = 44.0 \text{ db}$ (at $f = 5400 \text{ Mc}$). Also from Reference 2, the receiver line loss is 0.75 db . We presume that the FPS-16 incorporates a gaseous discharge TR box with an insertion loss not in excess of 0.25 db . Thus, we assume that the receiver losses will not exceed $L_R = 1.0 \text{ db}$.

* Interesting corroborative evidence has been afforded by Reference 14, from several air to air tracks of the Falcon GAR-9 and GAR-11 X band missiles. The vehicle antenna was circularly polarized with 6 db directivity at 180° aspect. The pattern was broad and flat, with 3 db points at 60° off axis. While tracking at 180° aspect, in all initial tests complete signal drop out (flame attenuation exceeded the 40 db AGC limit) occurred within 1/2 second of the engine start time, and signal recovery occurred immediately after engine burn-out.

The actual received power anticipated at the preamplifier input, therefore is:

	Site A		Site B		Site C	
	Gains	Losses	Gains	Losses	Gains	Losses
P_T	27.8 dbw		27.8 dbw		27.8 dbw	
L_T		1.5 db		1.5 db		1.5 db
G_T		6.0 db	0.0 db		3.4 db	
L_P		182.0 db		144.4 db		145.2 db
G_R	44.0 db		44.0 db		44.0 db	
L_R		1.0 db		1.0 db		1.0 db
		-118.7 dbw		-75.1 dbw		-72.5 dbw

and the actual received power anticipated at the antenna terminals is P_{R_a} ,

P_{R_a}	-117.7 dbw	-74.1 dbw	-71.5 dbw
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From Reference 2, the noise figure of the receiver with a conventional mixer is stated not to exceed 11 db, which corresponds to an absolute noise factor of 12.6, or to a receiver system noise temperature of 3364°K , which, in turn, corresponds to $T = 35.3$ db above 1°K .

From Reference 2, the receiver bandwidth is selected automatically to be 8 Mc, or $B = 69.0$ db above 1 cps.

Reference 2 notes that although the FPS-16 usually will track down to a S/N of -3 db, smooth tracking requires a S/N of 10 db or better, high precision tracking requires a S/N of about 20 db, and maximum accuracy tracking requires a S/N of 30 db. In addition, WSMR representatives stated (Reference 12) that they would require that $S/N \geq 20$ db for ATHENA operations.

The required received power at the antenna terminals, therefore, is

k	-228.6 dbw
T	35.3 db
B	69.0 db
S/N	20.0 db
<hr/>	
P_{R_r}	-104.3 dbw

and the signal margins for the three sites are:

Signal Margins: -13.4 db, for site A,
34.2 db, for site B,
32.8 db, for site C.

It should be noted that an operational mode requiring receiver antenna polarization change from vertical to horizontal (as the polarization misalignment angle passes through 45°) is undesirable. If the circular polarization modification kit should be available, it should be incorporated at sites B and C. In this case, we would include an additional linear-to-circular polarization loss of 3 db, which we can well afford from the signal margins for sites B and C.

Circular polarization reception at site A would correspond to a transmitter antenna directivity of $G_T = -2$ db and a receiver antenna directivity of $G_R = 44.0 - 3.0 = 41.0$ db, thus improving the system gain by 1 db.

Replacement of the FPS-16 12' reflector by a 16' reflector would provide an increase in gain of only 2.5 db.

Modification of the conventional receiver to include a parametric amplifier would reduce the receiver noise figure from 11 db to 5 db. This would change the effective receiver system noise temperature from 3364°K to 626°K , and improve the system gain by 7.3 db.

3.3 CONCLUSIONS

The results of the foregoing analysis were presented at the Reference 13 Conference. At that conference it was established by WSMR that the launch-area radars will be located in the plane of the trajectory, Site A. Based on this decision, the following conclusions were reached:

- a. The beacon antenna must be designed to provide a minimum gain of 4 db at 180° aspect, including polarization misalignment. Dropouts below 4 db must not occur for more than 1/4 second due to vehicle roll.
- b. Both FPS-16 radars will have parametric amplifiers.
- c. The beacon tracking link will be designed to operate to the parameters noted below.

	Gains	Losses	
P_T	26.0 dbw		(400 watts)
L_T		1.5 db	
G_T	4.0 db		(increased directivity)
L_P		182.0 db	
G_R	44.0 db		
<hr/>			
P_{R_a}		-109.5 dbw	
k		-228.6 dbw	
T	28.0 db		(parametric amplifier)
B	69.0 db		
S/N	12.0 db		(lowered threshold)
<hr/>			
P_{R_r}		-119.6 dbw	

Signal margin ≈ 10 db.

4.0 TRACKING LINK CALCULATIONS

4.1 SECOND STAGE BEACON TRACKING

Having determined that the second stage beacon antenna must have at least 4 db gain at tail aspect, we conclude that its pattern should be as indicated by Figure 6, as a minimum.

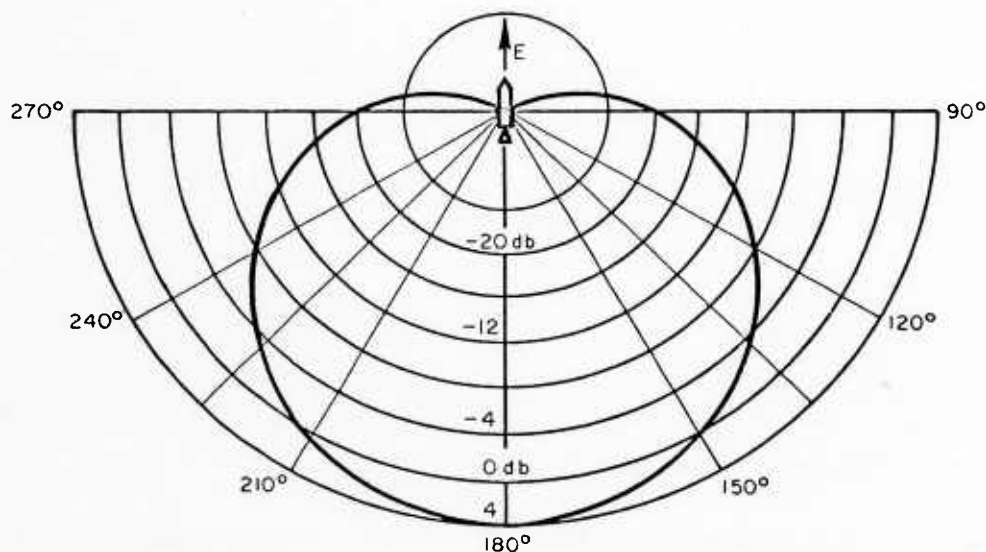


Figure 6. Antenna Pattern, Second Stage Beacon

4.1.1 Launch Site to Termination of Second Thrust

The second stage beacon can be tracked from the launch site with a 10 db signal margin, through the termination of second stage propulsion, under the assumptions of paragraph 3.3.

4.1.2 Either Site to Maximum Range

After the termination of second stage propulsion, flame attenuation will cease to be a problem. Therefore, it is of interest to determine the maximum range to which we may expect to track.

As in paragraph 3.3, the required received power is $P_{R_r} = -119.6$ dbw.

Above this threshold of receiver sensitivity, we wish to allow a signal margin of 5 db, so we require the actual received power to be $P_{R_a} = 114.6$ dbw.

Assuming the antenna pattern to be that of Figure 6, the antenna directivity at side aspect is -20 db. The "power budget", then, is:

	Gains	Losses
P_T	26.0 dbw	
L_T		1.5 db
G_T		20.0 db
L_P		
G_R	44.0 db	
<hr/>		
P_{R_a}		-114.6 dbw,

from which $L_P = -163.1$ db. Since $L_P = R^2 f^2$ (96.6 db) and $f^2 = 14.6$ db, $R^2 = 51.9$ db = 155,000 miles, and $R = 390$ miles.

We conclude that the second stage beacon can be tracked from the launch site down to the horizon.

The expended second stage will impact about 307 miles downrange from the launch site, i.e., not more than 110 miles uprange from the most distant radar at WSMR. For this range ($R = 110$ miles), $R^2 = 12100 = 40.8$ db, and the path loss would be $L_P = -152.0$ db. Reconstructing the "power budget" above, we see that we could tolerate an antenna gain as low as $G_T = -31.1$ db, i.e., if the antenna directivity were no worse than -31 db for the forward aspects (270° to 90° , including the nose), we should expect to be able to track the second stage beacon from WSMR, down to the horizon.

4.2 FOURTH STAGE BEACON TRACKING - MAXIMUM RANGE

As before, $P_{R_r} = -119.6$ dbw and $P_{R_a} = -114.6$ dbw.

We assume that the fourth stage beacon antenna will have an omnidirectional ($E\theta$) pattern and a minimum directivity of $G_T = -10.0$ db. The "power budget", then is:

	Gains	Losses
P_T	26.0 dbw	
L_T		1.5 db
G_T		10.0 db
L_P		
G_R	44.0 db	
<hr/>		
P_{R_a}		-114.6 dbw,

from which $L_P = -173.1$ db. The fourth stage will re-enter the atmosphere at hypersonic velocity, so that we must account for a certain amount of plasma sheath attenuation. Time has not permitted an accurate calculation of this attenuation, but 10 db appears to be a conservative estimate. This leaves 163.1 db for "free space" loss, from which the maximum range (as in paragraph 4.1.2) is 390 miles.

4.3 RADAR (SKIN) TRACKING

4.3.1 Launch Site to Termination of Second Thrust

In order to avoid ambiguities in the nomenclature, we deduce the (monostatic) radar equation, as follows:

P_T = transmitted power,

R = range of target,

G = radar antenna directivity,

σ = effective area of target,

A = effective area of receiving antenna,

L_T = transmitter losses (transmission line),

L_R = receiver losses (transmission line and diplexer),

L_P = (two-way) path loss.

$\frac{P_T}{4\pi R^2}$ = power density on the surface of a sphere of radius R , centered at the transmitter and passing through the target, assuming an isotropic radiator.

$\frac{P_T G}{4\pi R^2}$ = power density at the target, assuming a directional antenna.

$\frac{P_T G}{4\pi R^2} \sigma$ = total power which irradiates the target

$\frac{P_T G}{4\pi R^2} \cdot \frac{\sigma}{4\pi R^2}$ = power density on the surface of a sphere of radius R , centered at the target and passing through the receiving antenna, assuming that all the energy impinging upon the target is reradiated isotropically

$\frac{P_T G}{4\pi R^2} \cdot \frac{\sigma}{4\pi R^2} A$ = total power received at radar antenna

$$= \frac{P_T G}{4\pi R^2} \frac{\sigma}{4\pi R^2} \frac{G\lambda^2}{4\pi}$$

Thus, the actual received power (incorporating line losses), is given by

$$P_{R_a} = \frac{P_T G^2}{L_T L_P L_R},$$

where

$$L_P = \text{"free space" loss} = \frac{(4\pi)^3 R^4}{\lambda^2 \sigma}$$

As before, the required signal power, P_{R_r} , is given by $P_{R_r} = k T B S/N$, and the signal margin is $P_{R_a} - P_{R_r}$. The transmitter output power of the FPS-16 is one megawatt, so $P_T = 60.0$ dbw.

We assume that the transmitter losses will not exceed $L_T = 1.5$ db.

As before, $G = 44.0$ db, so $G^2 = 88.0$ db.

To determine the path loss, we note that $R^2 = 2384$, as in paragraphs 3.1 and 4.1.1. The carrier frequency is 5400 Mc. From Reference 15, the effective echoing area of diffuse (soft) reflection from the rocket exhaust at tail aspect is assumed to be $\sigma = 0.4$ meters². Thus, the total path loss is $L_P = 257.9$ db.

	Gains	Losses
P_T	60.0 dbw	
G^2	88.0 db	
L_T		1.5 db
L_P		257.9 db
<hr/>		
P_{R_a}		-111.4 dbw (at the antenna terminals)

Assume for the moment a receiver with a conventional mixer, and an effective system noise temperature of 3364°K, which corresponds to $T = 35.3$ db. The receiver bandwidth for radar (skin) tracking (Reference 2) is 2 Mc, so $B = 63.0$ db. As before, we assume that $S/N = 12.0$ db.

	Gains	Losses
k		228.6 dbw
T	35.3 db	
B	63.0 db	
S/N	12.0 db	
<hr/>		
P_{R_r}		-118.3 dbw

Thus, the signal margin is 6.9 db.

Normally, this margin would suffice to certify the link as acceptable. The calculation, however, is based upon one subjective parameter, viz., the "best guess" of $\sigma = 0.4 \text{ meters}^2$. Should the effective echoing area actually turn out to be as small as $\sigma = 0.1 \text{ meters}^2$, the signal margin would be reduced by 6 db.

The link is certified as acceptable, therefore, only with the inclusion of a parametric amplifier in the receiver system.

4.3.2 Either Site to Maximum Range

Having determined that skin tracking is feasible through the termination of second stage propulsion, we estimate the maximum range.

Assuming again the parametric amplifier, the required receiver power is given by

	Gains	Losses
k		228.6 dbw
T	28.0 db	
B	63.0 db	
S/N	12.0 db	
P_{R_r}		-125.6 dbw

Above this threshold of receiver sensitivity, we wish to allow a signal margin of 5 db, so we require the actual received power to be $P_{R_a} = -120.6 \text{ dbw}$, from which $L_P = 267.1 \text{ db}$.

We determine the maximum range from $L_P = \frac{(4\pi)^3 R^4}{\lambda^2 \sigma} = 267.1 \text{ db}$, where we must first estimate the effective echoing area.

The nose aspect is the least favorable and presents, approximately, a hemispherical cap of diameter 2 feet. Since the radar frequency is

5400 Mc, the wavelength is $\lambda = 0.182$ feet. Since the wavelength is less than the hemispherical radius, r , the effective echoing area is approximated by $\sigma \simeq \pi r^2 = \pi \text{ feet}^2$.

Thus,

$$L_P = 267.1 \text{ db} = \frac{64\pi^3 R^4}{\lambda^2 \sigma},$$

$$R^4 \simeq \frac{\lambda^2}{64\pi^2} \cdot 10^{26.7} \text{ feet}^4,$$

$$R \simeq 77 \text{ miles}.$$

When the target is viewed from the launch site, a less pessimistic value for effective echoing area is justified. Subjectively, we should probably increase σ by a factor of 4, which would increase R^2 by a factor of 2, and increase R to something like 100 miles.

On the other hand, as we view the target from the WSMR sites, we should allow something of the order of 20 db for two-way attenuation through the hypersonic plasma sheath. Thus, we would allocate only 247 db to "free space" loss, whence

$$R^4 \simeq \frac{\lambda^2}{64\pi^2} \cdot 10^{24.7} \text{ feet}^4,$$

$$R \simeq 24 \text{ miles}.$$

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12. Conference on Radar Siting, Antenna Patterns and Flame Attenuation, at AMR on 24 January 1963, including the following participants:

E. Fetner,	AMR (RCA-MTP)
A. E. Hoffman-Heyden,	AMR (RCA-MTP)
P. S. White,	WSMR
Lt. J. McSherry,	AFMDC
E. H. Serveson,	Aerospace
L. R. Norwood,	Aerospace

Air Force Report No.
BSD-TDR-63-144

Report No.
TDR-169(3153-01)TN-6

13. Conference on Radar Siting, Antenna Patterns and Flame Attenuation, at WSMR on 13 February 1963, including the following participants:

C. Bustamante,	WSMR
J. S. Marsh,	WSMR
P. S. White,	WSMR
Lt. J. McSherry,	AFMDC
Lt. Col. Parker,	BSD
E. H. Serveson,	Aerospace
L. R. Norwood,	Aerospace

14. Personal communication with V. T. Norwood, Hughes Aircraft Company.
15. Personal communication with A. E. Hoffman-Heyden, RCA-MTP, estimate from AMR data.

UNCLASSIFIED	<p>Aerospace Corporation, El Segundo, California. AN ANALYSIS OF ATHENA TRACKING PROBLEMS (THE ATHENA PORTION OF ABRES 627a) (UNCLASSIFIED TITLE), prepared by Lawrence Norwood. November 1963. [43]p. incl. illus. (Report TDR-169(3153-01)TN-6; BSD-TDR-63-144) (Contract AF 04(695)-169)</p> <p>The ATHENA program has been conceived to provide a systematic method for investigation of atmospheric reentry phenomena and, through the use of a test vehicle especially designed to reproduce ICBM trajectory dynamics, to facilitate the evaluation of advanced reentry concepts. The White Sands Missile Range has been selected to make critical reentry measurements. Green River, Utah, will be the launch site. Tracking radar will be provided at the launch site to perform flight safety surveillance, to provide acquisition data for range radars, and to monitor (over)</p>
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